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Office, Program Planning and Evaluation

INTRODUCTION

Before discussing spacecraft propulsion, it is well to define what we mean by a spacecraft. A spacecraft is a vehicle that travels outside the earth's atmosphere for the greater part of its mission. Using this definition, the simplest examples of spacecraft are high-altitude sounding rockets or long-range ballistic missiles, popularly known as IRBM's (intermediate range ballistic missiles) and ICBM's (intercontinental ballistic missiles). From these we go to the more advanced devices - the earth satellite, the deep space probe, the interplanetary vehicle, and so forth. The research and development of the missiles was largely responsible for providing rocket propulsion systems adequate for advanced space activities.

Using missiles as an example of a simple spacecraft, figure 1 shows that to cause a body to travel from point A on the earth's surface to point B on the earth's surface by means of a ballistic trajectory in space requires that the body be accelerated to a velocity of 9500 miles per hour if A and B are 1500 miles apart and a velocity of 16,000 miles per hour if A and B are 5500 miles apart. In each case, the accelerating period should be quite short, being measured in minutes. The relation between distance traveled and the velocity to which the vehicle is accelerated is not linear as is shown in figure 2. Figure 2 also shows that the distance becomes asymptotic to a velocity of about 17,000 miles per hour. It is assumed in the figure that the angle of flight relative to the earth's surface at the end of the accelerating period is such so as to give maximum range. This angle varies from about 40° to the horizontal for the 1500 mile range to about 25° for the 5500 mile range. At the 17,000 miles per hour, if the vehicle is above the earth's atmosphere and at an angle reasonably close to tangency with the earth's surface, the distance becomes infinite, that is, the vehicle goes into orbit around the earth. In this case the velocity imparted to the body at the appropriate distance from the earth (about 300 miles) produces a centrifugal force just equal to the earth's gravitational pull.

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[1]Much of the information contained in this paper has been published in two previous papers by the author: (1) "Spacecraft Propulsion and High Energy Fuels," presented at the Symposium on High Energy Fuels of the Meeting of the American Chemical Society in Boston, Massachusetts, April 10, 1959; and (2) "Aircraft and Spacecraft Propulsion," presented at the Special Anniversary Meeting of the Canadian Aeronautical Institute, February 23-24, 1959 in Montreal, Canada.

Going a step farther (fig. 3), if the velocity of the vehicle in orbit is increased an additional 7000 miles per hour, it will travel outward from the earth orbit and if aimed correctly, will approach the moon. At about 200,000 miles from the earth and 40,000 miles from the moon, the moon's gravitational pull on the body will exceed that of the earth. If, as the vehicle approaches the moon, it is slowed down by about 1500 miles per hour, it will go into orbit around the moon. An additional decrease in velocity of 4000 miles an hour and the body will be drawn into and land on the moon. The velocities given here will vary with the specific flight plan chosen.

Repeating these velocities in magnitude but in the opposite direction and with appropriate guidance, the vehicle will go into moon orbit, leave the moon orbit, and start toward the earth, enter an earth orbit, and finally, land on the earth. The dimensions in figure 3 are approximately to scale.

We have seen from these examples that controlled space travel is largely a matter of imparting velocity changes to the vehicle, changes that vary both in magnitude and direction. It is noticed that the actual velocity of the vehicle is not necessarily stated, only the velocity change. For instance, although the vehicle left the earth orbit at a velocity of 24,000 miles per hour (17,000 + 7000), its velocity relative to the earth continually decreased as the moving vehicle did work against the earth's gravitational field. At 10,000 miles from the earth its velocity relative to the earth would be about 13,000 miles per hour, and the velocity would continue to decrease until the gravitational pull of the moon caused the vehicle to accelerate again.

We now define the spacecraft propulsion system as a device for changing the velocity of a spacecraft in either magnitude or direction. Referring again to the moon landing and return, the total velocity change required is 59,000 miles per hour (table I). Of this velocity change, 42,000 miles per hour must be supplied by the propulsion system. For the final 17,000 miles per hour increment to return from earth orbit to the earth, all but a small fraction will be accomplished by means of aerodynamic drag as the vehicle passes through the earth's atmosphere.

Representative velocity changes for different missions as estimated by Sutton are given in figure 4. These velocities will vary with the particular mission path, but not sufficiently to change the basic picture. Where a return to earth is involved, the velocity decrease of about 17,000 miles per hour from earth orbit to earth's surface is for the most part accomplished by the resistance of the earth's atmosphere.

To change the velocity of a spacecraft in magnitude or direction, the propulsion system must exert a force on the body. To exert a force artificially on a body in space, a mass must be accelerated from, or energy must be discharged from the body, thus producing a reaction force on the body; or mass or energy must strike the body, adhering to, or being reflected from the body. The system of accelerating a mass from the body is the one currently used.

The mass accelerated from the body can be either solid, liquid, or gas. As a matter of convenience, it is accelerated in the form of a gas. This gas is termed the propellant. Reflection of photons (energy) from the sun off the body is under research consideration but will be referred to only briefly here.

Through this discussion, mass will be used, rather than weight. The weight of a body is the gravitational pull of the earth on the body, and, consequently, varies inversely as the square of the distance from the earth's center. The mass of the body is an inherent property of the body and, under the conditions discussed in this paper, remains constant. The weight of the body at earth's sea level is numerically equal to the mass of the body.

We know that if the propellant is discharged at a given rate having been accelerated within the vehicle to a given velocity, the force produced on the vehicle is expressed by

$$F = \frac{M_P}{t} V_P \tag{1}$$

in which

F force produced

M_P mass of propellant discharged

t time during which discharge takes place (M_P/t is therefore rate of propellant discharge)

V_P velocity of propellant discharge relative to the vehicle, that is, velocity to which propellant is accelerated

The change in velocity produced on the vehicle is

$$V_V = \frac{F}{M_V} t \tag{2}$$

in which

M_V mass of the vehicle

V_V the <u>change</u> in vehicle velocity produced by the force F

t the time during which the force acts

The time during which the force acts is the time during which the propellant is discharged. The value of force F in equation (1) can be substituted in equation (2), giving

$$V_V \approx \frac{M_P}{M_V} V_P \tag{3}$$

We see that we have a relation involving two velocities and two masses. Since in practice, the mass M_V of the vehicle at the start of velocity change must have added to it the mass of the propellant carried; and, since this propellant mass aboard the vehicle is progressively decreased during the accelerating period, equation (3) should be rewritten as

$$V_V = f\left(\frac{M_V + M_P}{M_V}\right)V_P \tag{4}$$

It can be shown that the function of $\dfrac{M_V}{M_V + M_P}$ **is in fact,**

$$\ln \frac{M_V + M_P}{M_V}$$

the natural logarithm of the ratio of the mass of the vehicle plus propellant at the start of acceleration to the mass of the vehicle less propellant discharged during the acceleration.

We now rewrite equation (4) as

$$V_V = V_P \ln\left(\frac{M_G}{M_V}\right) \tag{5}$$

in which

$$M_G = M_V + M_P$$

The reason the logarithm enters into the relation is that each increment of propellant carried aboard must be accelerated as part of the vehicle until such time as the increment is discharged. To give a general idea of the values of the logarithms for the range of interest, table II is presented. We see that an eightfold increase in the ratio of masses results in approximately a threefold increase in the logarithm and therefore in the vehicle velocity. The velocity increase V_V is added vectorily to any velocity the vehicle had at the start of the accelerating period.

The importance of equation (5) cannot be overemphasized. The equation shows that the velocity change to the vehicle varies directly with the velocity with which the propellant is discharged. For this reason much of the present discussion will be on the means of obtaining higher propellant velocities. It is for this reason that we will go in our discussion from chemical thermal rockets, to nuclear thermal rockets, to nuclear electric rockets.

Equation (5) assumes there are no other forces acting on the vehicle other than the propulsive force. In practice there are always gravitational forces acting on the vehicle. The magnitude and direction of these forces depends on the proximity of the vehicle to the different heavenly bodies. Launching from the surface of the earth for instance, subjects the vehicle to the earth's full gravitational field. In cases where the gravitational forces are significant a modification is required to equation (5) as follows:

$$V_V = V_P \ln \frac{M_G}{M_V} - g_e t \tag{6}$$

in which

g_e average gravitational force acting on vehicle during the accelerating period

t time for acceleration

Whether or not the term $g_e t$ need be considered depends on its magnitude in relation to $V_P \ln\left(\frac{M_G}{M_V}\right)$. The velocity of a vehicle in space is of course always changing because of the vector sum of all the gravitational forces acting on it. It was this fact that led astronomers to locate the planets Neptune and Pluto through their gravitational effects on the velocities of the other planets.

One more change will be made in equation (5). In space flight the vehicle is often divided into two major parts. The first is the propulsion system, propellant tankage, and necessary additional structure. The second is the device that is performing the desired operation in space. This part is termed the payload. Unfortunately, payload cannot be defined precisely. On current spacecraft, the payload is that portion that contains the instruments or passengers, which are being used in accomplishing the specific objective of the mission. From this standpoint the vehicle mass can be separated into two parts

$$M_V = M_{St} + M_{PL}$$

in which

M_{St} the mass of propulsion system, propellant tanks, additional structure

M_{PL} the mass of the payload that performs the actual space mission

In a multistaged rocket, the payload of each stage consists of all the subsequent stages, plus the final payload. Equation (6) can now be rewritten

$$V_V = V_P \ln \frac{1}{\frac{M_{St}}{M_G} + \frac{M_{PL}}{M_G}} - g_e t \tag{7}$$

in which

$$M_G = M_{St} + M_P + M_{PL} \tag{8}$$

Equation (7) indicates the importance of a low value of M_{St}, that is, of the ratio M_{St}/M_G.

We emphasized previously the importance of propellant velocity. We will now examine the effect of the ratio M_{St}/M_G. To do this we must assume representative values. In a chemical rocket a ratio M_{St}/M_G of 0.10 is reasonably representative. Values of M_{PL}/M_G from 0.05 to 0.33 are appropriate. Consider first the 0.05 value. A decrease of 20 percent in M_{St}/M_G from 0.10 to 0.08 changes the fraction $\dfrac{1}{\dfrac{M_{St}}{M_G} + \dfrac{M_{PL}}{M_G}}$ from 6.67 to 7.69 and the natural logarithm of the fraction from 1.90 to 2.04, giving an increase of 7.5 percent in the vehicle velocity. Using the value of 0.33 for M_{PL}/M_G and the same values for M_{St}/M_G, the increase in vehicle velocity becomes 6 percent. Although the effects here are less than those with changes in propellant velocity, they are still important.

We have now presented the basic equations that determine the change in vehicle velocity as a result of the force produced by the propulsion system. We see that the factors of major importance in obtaining a given payload velocity change are: (1) the velocity with which the propellant is ejected from the vehicle by the propulsion system, (2) the ratio of the weight of the weight of the vehicle less propellant and payload to the gross weight, and (3) (if significant gravitational forces are acting on the vehicle), the time during which the force acts.

One more term will be discussed before taking up the propulsion system - that is, specific impulse. Equation (7) and the preceding equations use the propellant velocity V_P as the significant variable. The term more generally used is <u>specific impulse</u>, I_{Sp}, that is, the pounds of thrust (force) produced by the propulsion system for each pound per second of propellant discharged. The propellant velocity in feet per second is, in fact, numerically the force produced by the propellant in poundals. Therefore, dividing the propellant velocity in feet per second by g (32.2 ft sec^{-2}) gives the propellant force in pounds of thrust per pound of propellant discharged per second; or expressed generally,

$$V_P = g\ I_{Sp} \tag{9}$$

We are now ready to turn to the propulsion system.

The propellant system, as was stated, is a device that produces a force by accelerating a mass (propellant) from the vehicle. Therefore, the system must contain in addition to the propellant, an energy source, a device for converting this energy into a form (heat or electricity) that can be used to accelerate the propellant and also a means for accelerating the propellant. The propulsion system can be conveniently outlined as shown in figure 5. The energy source can be nuclear or chemical. The propellant can be carried in the form of a solid, liquid, or gas. In practice, it is carried as a solid or liquid and for convenience, converted into a gas before being accelerated from the vehicle. The propellant could be energy (photons) rather than mass. The powerplant performs the three functions listed. There are the materials of which the powerplant is made, and in some cases, a heat-transfer fluid is required to transfer energy in the form of heat within the powerplant.

powerplant or to cool the powerplant. The propulsion system will be discussed under these five major headings.

The space propulsion systems are, in general, referred to as rockets. They can be divided into thermal or electric rockets. The thermal rockets are either chemical or nuclear (fig. 6). In the chemical thermal rocket, the combustion of fuel and oxidant form a hot combustion gas (propellant), which is accelerated by expansion through the nozzle. In the nuclear thermal rocket the gaseous propellant is passed through a nuclear reactor and thereby, heated. The hot propellant is expanded through the nozzle and so accelerated.

The nuclear electric rocket is shown in figure 7. It, as well as the thermal rockets, will be discussed in more detail later. It is sufficient to say for the present that with the electric rocket electric power from an electric power generating system accelerates the propellant.

With this brief description of the systems, we will turn to a detailed discussion of the spacecraft propulsion system components as diagrammed in figure 5.

ENERGY SOURCE AND PROPELLANTS

Chemical and Nuclear Reactions

Energy from the source, either chemical or nuclear, is transferred into heat or electricity by either combination of two or more elements or compounds or by decomposition of one or more elements or compounds (fig. 8). Chemical energy can be converted into heat or into electricity, the basic process is the same - interchange of electrons between the elements or compounds involved. Nuclear energy can be converted directly into heat and from heat to electricity either by means of conventional electric power generating devices or by means of thermopiles, or considering solar energy directly into electricity through photoelectric cells. These latter processes are currently of too low efficiency to be used as propulsion devices. Research to improve these efficiencies is being pursued actively. The nuclear energy conversion to heat can take place in a nuclear reactor or can be from the sun in the form of radiant energy. For space propulsion all phases listed in figure 8 are under consideration, although current usage is limited to chemical combination or decomposition with the release of heat.

It is advisable to discuss briefly the difference between a chemical and a nuclear reaction. Figure 9 shows examples of simple chemical and nuclear reactions. In each case, the nucleus of the atom is represented by the solid circle. The protons in the nucleus are indicated by +'s and the neutrons by n's. The electron orbits surrounding the nucleus are indicated by the dashed circles and the electrons by the solid circles placed on the orbits. It is noted that in the chemical reaction the nucleii remained unchanged, only the electrons being involved. The elements are therefore unchanged, only rearranged. In the nuclear combination, heavy hydrogen (also called deuterium) is shown. In this case the nucleus has one neutron as well as the proton. Now, in the reaction, the nucleii are changed. The two heavy hydrogen atoms become one helium atom plus a free neutron. As will be shown later, this difference in the chemical and nuclear reactions results in a tremendous difference in energy release per unit mass of material involved.

Figure 10 shows examples of chemical and nuclear combination and dissociation using the more conventional symbols for the chemical elements involved. The symbol eV indicates the energy released in the reaction. We will discuss its magnitude later. In the nuclear equations, the left-hand subscript indicates the number of protons in the nucleus (that is, the atomic number). The right-hand subscript indicates the number of protons plus the number of neutrons (that is, the atomic mass). The third nuclear equation represents radioactive decay such as occurs in nature.

The energy source in the space propulsion system will consist of one or more of the chemical elements as listed in the periodic table (fig. 11). The elements may be in atomic or molecular form. For chemical reaction, molecules rather than atoms are used. Chemical energy conversion to heat or electricity will be considered first.

Chemical Energy Conversion

A chemical reaction involves the atomic electrons and not the nucleus. For the most part the electrons involved are those in the outer shell. These vary progressively in number from one to eight. These are shown in figure 11 diagrammatically around the Roman numerals designating the respective columns. An enlargement for four of the elements is shown in figure 12, which includes two elements with one electron in the outer shell, indicated by K for hydrogen (H) and P for cesium (Cs). The two other elements shown, oxygen (O) and polonium (Po), have six electrons in the outer shell. Since the number of electrons involved in the outer shells varies progressively from one to eight and since these are the electrons in general that are involved in a chemical reaction, it is reasonable to expect that a curve of energy release versus atomic number should be periodic in form and that the different maximum energies in the curve should be about the same. The proof of this is shown in figure 13 in which the calculated energy release, either in the form of heat or electricity, is shown for the various elements reacting with oxygen. In choosing fuels and oxidants, the energy release per unit mass is of more interest than the release per molecule formed. The data in figure 13 are replotted on a mass basis in figure 14. Since the molecules formed in the reaction become heavier as the atomic number of the element increases and since the ratio of "fuel" to "oxygen" in the reaction varies, the curve shows continually decreasing maxima as atomic number is increased and there are certain changes in the elements from figure 13 to 14 at which the maxima occur. Figure 14 indicates that the elements of interest as chemical rocket fuels are hydrogen and those of atomic numbers close to beryllium (Be) and aluminum (Al) and possibly scandium (Sc). This section of the curve is enlarged in figure 15, in which the primary oxides and the liquefaction (boiling) temperature of certain of the oxides are also shown. The oxides listed are in fact ceramics. Without going into detail, it is reasonable to state that high liquefaction temperature combustion products will probably be a source of trouble, from the standpoint of extracting the heat both from the liquefied or solidified drops or from deposits formed within the propulsion system. For this reason, the light metals will probably be used in rocket fuels in reasonably limited percentages.

From figure 15 the elements of interest as space propulsion fuels in chemical-thermal systems are, in general, those listed under heat in figure 16. Without going into detail, the oxidants of most interest are currently fluorine and oxygen. Two elements, nitrogen (N) and chlorine (Cl), are listed as carriers. Although the term carrier is not too satisfactory, these two elements are used extensively to form suitable solid or liquid rocket fuels and oxidants. The lower half of figure 16 lists certain compounds that are used in the direct production of electricity by chemical reaction. The corresponding numbers under A and B represent corresponding half cells or as they might be termed, oxidants and fuels. With the exception of the oxygen-hydrogen cell, the elements involved have high atomic weights and, therefore, yield low energy outputs per mass of reactants. Direct conversion of chemical energy to electricity is not currently of interest as a means of spacecraft propulsion, but is of interest as a means of producing electric power for other use aboard the craft.

Following the conversion of chemical energy into heat within the propulsion system powerplant the heat must be transferred to the propellant. Since, in this case the propellant is the exhaust gas formed during the combustion of fuel and oxidant, the first two steps, conversion and transfer, take place simultaneously. The acceleration of the propellant is accomplished by expansion through the rocket nozzle. The process may be visualized as follows: According to the kinetic theory of gases, the kinetic energy of a molecule is dependent on the temperature. In other words, temperature is a measure of molecular kinetic energy

$$\frac{1}{2} mV_m^2 = KT \tag{10}$$

in which

m mean molecular weight of exhaust gas

V_m mean molecular velocity

K a constant (varying somewhat with the molecules involved)

T_c absolute combustion temperature

Equation (10) may be rewritten

$$V_m = \sqrt{\frac{KT_c}{m}}$$

$$\sim \sqrt{\frac{T_c}{m}} \tag{11}$$

which states that the mean molecular velocity varies as the square root of the gas temperature divided by the mean molecular weight. The mean molecular velocity, V_m, is, in fact, a measure of the heat (more correctly the square root of the heat energy) released per unit mass as expressed in figures 13 to 15. The velocity V_m is a random velocity. The purpose of the expansion of the propellant gas through the rocket nozzle is to change this random velocity in part into a directed velocity, V_P, of the propellant as a whole. It can be shown that the velocity to which the propellant is accelerated is directly proportional to the random molecular velocity resulting from the combustion temperature; that is,

$$\begin{aligned} V_P &\sim V_m \\ &\sim \sqrt{\frac{T_c}{m}} \end{aligned} \tag{12}$$

In this manner the thermal energy of combustion expressed to a first approximation in figure 15 accelerates the propellant (combustion or exhaust gas). It is seen from equation (12) that a maximum value of T_c/m is desired; therefore, a high combustion temperature and a low mean molecular weight of combustion products is desired.

Since the propellant is the combustion product of fuel and oxidant, the values of T_c and m are determined by choice of fuel and oxidants. The combustion temperatures experienced in practice do not vary much - say, from 4500° to 7000° F, the square root of the ratio of absolute values being 1.22, which represents approximately the variation in propellant velocity or specific impulse as a result of temperature change. The variation in molecular weight, as will be shown later, is from about 32 to 11, the square root of the ratio being about 1.70. The total variation in specific impulse with currently considered fuels and oxidants is the product of these two numbers or about 2.0.

Since the maximum value of the ratio T_c/m is desired, the ratio of oxidant to fuel is adjusted to give this maximum, as shown in figure 17 for mixtures of oxygen and a boron hydride. It is seen that as the oxidant - fuel ratio is increased beyond the stoichiometric value of 3.5, the greater affinity of boron for oxygen results in hydrogen appearing in increasing amounts in the combustion products. Consequently, the average molecular weight continually decreases, and the maximum value of propellant velocity occurs at less than the maximum temperature.

The second major factor affecting specific impulse is the pressure ratio across the discharge nozzle. Considering the pressure ratios realized in practice and the limitations on discharge nozzle design, this factor is about 1.25 for the ratio of the specific impulse, or propellant velocity, of a rocket designed to operate outside the earth's atmosphere and one designed for sea-level operation.

Figure 18 lists the specific impulses of representative oxidants and fuels that are liquids or gases at atmospheric temperatures and pressures.

In each case, the oxidant is listed to the left (O_3 for example) and the fuel to the right (H_2 for example). It is noticed that as a generality, the simpler molecules have the higher specific impulse. Also, that with the larger molecules, hydrogen (a fuel) may appear in the oxidant and oxygen may appear in the fuel. This results in certain desirable characteristics at the expense of specific impulse. Those fuels (H_2) and oxidants (O_2, F_2 or O_3) that are normally gases are maintained as liquids by means of refrigeration and are termed cryogenics. The required liquefaction temperatures (boiling point) together with certain other properties of interest in evaluating the fuels and oxidants are shown in figure 19.

Solid chemical propellants consist either of an intermixed fuel and oxidant, in which case the term composite propellant is used or of one or more unstable compounds each of which will decompose of its own accord with the release of heat. Since many solid propellants in this second category are based largely on a colloid of nitroglycerin and nitrocellulose, they are often referred to as double-base propellants. For the composite solid propellants, the oxidant is usually a perchlorate or a nitrate. Again, the elements listed in figure 16 are desired.

Finally, so-called metastable, or free radical fuels and oxidants might be mentioned. These are essentially partially decomposed molecules that, because of their partial decomposition, yield higher heats of combustion than the more stable forms. Their inherent instability has so far precluded their use in chemical rocketry.

Nuclear Energy Conversion

The limitation of combustion temperature and combustion product molecular weight might be removed or lessened if an independent energy source is used to heat the propellant. Nuclear energy is of great interest from this standpoint. Following the same procedure used with chemical energy, we will first examine the heat output per unit weight of nuclear fuel consumed. Figure 20 shows the calculated energy release for nuclear fusion or fission. In the nuclear reaction the atomic nucleus, rather than the surrounding electrons, are involved. Therefore, there is no periodic variation involved as with the chemical reaction. Without going into the reasoning, the binding energy of the atomic nucleii is highest for the atoms of intermediate weight - say, those between calcium (Ca) of atomic mass number 40 and bromine (Br) of atomic mass number 80. For elements lighter than these, energy is generated through atomic fusion. For heavier elements energy is released through fission. The values shown are for fusion or fission to elements in the calcium to bromine range. The hydrogen fusion as obtained currently is to helium, rather than to - say, calcium. Consequently, the energy release is about 50×10^9 Btu per pound, rather than 335×10^9 shown. It is noted that these values are roughly 10^7 times those shown for the chemical reactions in figure 14. In practice, a ratio of 10^6 is more reasonable because of the low "combustion" efficiency of the nuclear reaction.

We see that with nuclear energy the mass of fuel consumed is negligible, but a propellant must be provided. Since the basic relation involved (eq. (12)) shows the desirability of a low molecular weight propellant, hydrogen is chosen. The molecular weight of hydrogen is 2, approximately one-sixth to one-sixteenth that proposed with the chemical fuels giving, at the same temperature, two to four times the specific impulse, that is, propellant velocity.

In the case of the nuclear thermal rocket, the propellant is passed through a nuclear reactor and thus heated. It is then expanded through a nozzle, and so accelerated as was the case with the chemical rocket. The temperature to which the propellant is heated is now limited by the temperature to which the reactor can be heated. This will be discussed in more detail later. Suffice it to say that current materials limit it to values lower than the combustion temperature in the chemical rockets.

In summary, with nuclear energy the elements are the same as those normally considered for nuclear reactors, the heavy elements, uranium or plutonium for fission reactors, and if the problems of controlled fusion are solved, the light elements, hydrogen and possibly lithium, for the fusion reactor. It is noted that the number of elements considered as energy sources, either chemical or nuclear, is relatively small.

Electrical Energy Conversion

The chemical or nuclear rockets discussed so far can be termed thermal rockets, since energy in the form of heat is used to accelerate the propellant. Because temperature is the major factor in determining heat, and because there is a limit to the temperature that can be withstood by the materials of which the rocket is made, it would be well to use a propellant acceleration process that does not use heat. The thermal rocket accelerates the propellant by first increasing the random velocity (heat) of the molecules and then by expansion through a nozzle, converting part of this random velocity to a directed velocity of the propellant as a whole. If the atoms, molecules, or even larger particles of propellant are charged electrically and placed within an electrostatic or electromagnetic field, the field voltage will impose a force on the charged particles and accelerate the propellant. In this case, the average acceleration of the individual particles becomes the acceleration of the propellant as a whole. The equations involved are similar to equation (12) for the thermal process:

$$\frac{1}{2}\, mV_m^2 \sim EQ \tag{13}$$

$$V_m \sim \sqrt{\frac{EQ}{m}} \tag{14}$$

in which

V_m average molecular velocity

E electrical charge imposed on the molecule m molecular (particle) weight

Q voltage imposed on the charged particles

The charge E is in general that provided by the removal of one electron from the particle. There is a limitation to the voltage that can be used, but it is sufficiently high that it need not be considered here. Therefore, since the value EQ in equation (14) is not limited as is T_c in equation (12), low molecular weight propellants are not required in the interests of high propellant velocities, that is, specific impulse. Since the temperature limitation has been removed or at least greatly lessened much higher specific impulses currently appear possible than in the case with thermal rockets.

An additional point must be considered, that is, the energy required to remove the electrons from the propellant particles and thus give the particles the necessary electric charge. This energy is lost to the system, that is, it does no useful work in accelerating the propellant. The energy required to remove one electron (the ionization energy) from each of the elements is shown in figure 21. The periodic variation in the curve is noted. It is also noted that the energy per pound required to ionize hydrogen is over 100 times the energy yield (fig. 15) from burning hydrogen with oxygen. Whereas, because of temperature limitation, light elements are desirable as propellants in thermal rockets, in electric rockets, heavier elements are desirable because of lower ionization energy requirements. The element cesium is an interesting possibility because the ionization energy is low and cesium is a liquid at ordinary temperatures.

In discussing the electric rockets we will look a little further into the matter of removing the electron (ionization). Figure 22 shows an oxygen atom on the left. It is noted that the number of electrons (8) equals the number of protons. Since each electron contains one negative charge and each proton contains one positive charge, the atom is electrically neutral. To the right is shown an oxygen ion. This is an atom that has an electron removed, indicated to the extreme right. Since the atom, which is now termed an ion, has one more proton than electron, it has a positive charge. The removed electron is, of course, negatively charged. Under normal conditions the electrons must be physically removed from the presence of the ions or they will recombine. The separation can be accomplished by having material present that will absorb the electrons and carry them from the ions. Having separated electrons and ions, each can be accelerated and discharged from the spacecraft by means of an electrostatic field. Again, the thrust produced is given by equation (1) in which M_p is now the sum of the mass of ions plus electrons. Actually, the mass of electrons discharged is so small compared with that of the ions that their effect can be neglected. This type of electrical system is often termed an <u>ion</u> jet or rocket.

If appropriate conditions of pressure and temperature exist, electrons and protons can remain together without recombining. In this case, an electromagnetic field is required to accelerate the propellant (mixture of ions and electrons). This type of electric system is termed a plasma jet or rocket and the mixture of ions and electrons is termed a <u>plasma</u>.

For the ion and plasma jets, nuclear energy will be used to produce the electric energy required for the ionization process and to produce the electrostatic or electromagnetic fields. Propellant velocities of several hundred thousand miles per hour (specific impulses of 10,000 or higher pounds thrust per pound of propellant discharged per second) can be produced with ion or plasma jets as compared with values of 6000 to 25,000 miles per hour with the thermal rockets.

Propellants

The propellants are summarized in figure 23. If the propellant is accelerated by means of the heat produced chemically, the propellant is the gas resulting from the combustion of fuel and oxidant. Remembering that the significant factor in determining propellant velocity is the ratio of propellant (combustion) temperature to propellant (combustion products) molecular weight, there is a limit to this ratio imposed by the temperature of combustion and the molecular weight of combustion products. The limit imposed on molecular weight can be partially removed by using nuclear energy as the source of heat. In this case the lightest molecular weight gas existing in nature, hydrogen, can be used as the propellant.

By using an electrostatic or electromagnetic field as the means of propellant acceleration, both temperature and molecular weight limitations are removed and propellant velocities of several hundred thousand miles per hour are possible. By going to energy discharge (photons) rather than mass, still higher specific impulses can be obtained, but the situation here is far from clear at the present time. Much research is needed.

It must be brought out that as the propellant velocity, that is, specific impulse, is increased, the rate of energy expenditure, that is, power in the propellant jet, is increased which itself presents limitations. This will be discussed further in a later section.

The phrase "of any element" in regard to plasmas indicates a lack of knowledge rather than a broad choice. The designation of "photons" under acceleration by radiation also indicates a need of research in order to specify more clearly profitable paths to follow.

In figure 24 representative specific impulses for the propellants are shown. All figures seem reasonably sure of attainment. For the higher values research and development are required. In addition to the specific impulse produced, the specific jet power which must be supplied is also indicated.

MATERIALS

We will turn now to the materials of which the powerplant is composed. Our treatment of them will be rather brief. Of the items listed in figure 5 we will consider only the metals and the ceramics. The material property most difficult to obtain in thermal powerplants is the ability to withstand the high operating temperatures desired. Practical turbine engines for aircraft were not achieved until materials were developed that could withstand stresses of the order of 25,000 pounds per square inch in the presence of hot oxidizing atmospheres. When alloys were developed that could operate under these conditions at temperatures of 1200° F, turbojet engines came into use. The progress made over a twelve-year period in this field is shown in figure 25. We see that during this period the operating temperature was increased from about 1350° to 1650° F - an increase of 300° F. These temperatures are appreciably below those desired for either chemical or nuclear rocket engines. Since the stresses and times of operation in spacecraft engines may be less than the case with the turbine aircraft engines, higher temperatures are feasible; but in any case the goal desired is some distance from realization. There are many highly competent people working in the field of high temperature materials. Progress in the field is difficult. Because of the inherent difficulties encountered, we cannot count on rapid advances. Research leading to a better understanding of the physics of solids may give us the improvements we want. To assist in understanding the problem, figure 26 is presented showing the variation in melting temperature of the elements as a function of atomic number. The melting temperature is the first criterion in determining the suitability of a material for high temperature operation. In general, the high stress operating temperature of an alloy is about two-thirds the melting temperature of the major ingredient. Since melting temperature is dependent on the arrangement of the electrons, a periodic variation is obtained with (other than the first maximum at carbon (C)) successively increasing maxima at silicon (Si), chromium (Cr), molybdenum (Mo), and tungsten (W). For the high temperature alloys we are interested in the metals occurring near these maxima. Specifically, grouped in the order of increasing desirability: (1) vanadium (V), chromium (Cr), iron (Fe), cobalt (Co), and nickel (Ni); (2) columbium (Nb), molybdenum (Mo), technetium (Tc); and (3) tantalum (Ta), tungsten (W), and Rhenium (Re). Each group is successively rarer in the earth's crust, which may or may not be objectionable, and is increasingly subject to corrosion in the presence of hot gases, which is objectionable. Each successive group is more difficult to fabricate. Much research is being conducted on alloys of these materials to solve these problems. The ceramics, including the oxides of the light metals previously referred to are also the subject of much research. Their low resistance to thermal shock is one of the difficult problems to overcome.

Figure 27 shows the material temperatures that will be achieved if temperatures of about 80 percent of the melting or sublimation temperature of the major constituent can be obtained. This figure may be considered to represent research goals.

HEAT-TRANSFER FLUIDS

Heat-transfer fluids are used in certain systems that require heat to be transferred from one part of the powerplant to another or that require cooling by other than direct radiation or conduction from the powerplant. Along with heat-transfer fluids, working fluids for closed turbine-drive systems can be considered. Properties desired are: high specific heat (i.e., low molecular weight), high density, low corrosivity, low response to radioactivity, an acceptable melting temperature. These properties are not mutually compatible. Figure 28 lists materials of particular interest.

In specific instances, such as using a liquid chemical rocket fuel or oxidant to cool the combustion chamber and the nozzle, the liquid available is determined primarily by its use as a fuel or oxidant. Representative values are shown in figure 29. In the figure the ordinate is Btu per second cooling capacity per pound of thrust produced. In each case, satisfactory use of the fuel as a coolant is reasonably assured, although chemical stability poses some problems. The use of oxygen or fluorine oxidants as coolants is also given consideration. As with hydrogen, the liquefied gases are being considered. Since the critical temperatures of oxygen and fluorine are somewhat high, a choice must be made as to whether or not the coolant is to be used above or below critical pressure. If the coolant is used below the critical pressure, it is limited by the boiling point. The solid portions of the oxidant bars (fig. 29) represent the heat capacities available within the limitations of the boiling points of the fluid at pressures normal for cooling. However, if higher pressures are used, for example 800 pounds per square inch, then the critical pressure is exceeded and there is no boiling point problem. Again, the engine wall provides the limit. The total heat capacity is represented by the total height of the bar for each oxidant.

THE POWERPLANT

We are now ready to discuss the powerplant as a unit. We stated in the Introduction that the powerplant performed three functions. It transforms energy, chemical or nuclear, to heat or electricity, transfers the energy to the propellant, and accelerates the propellant (fig. 30).

The chemical energy (fig. 31) is transformed into heat in a combustor and into electricity in a chemical battery. Nuclear energy is transformed into heat by means of a reactor, or a radioisotope source. If radiant energy from the sun is being used, it can either be converted into heat in the powerplant by means of a heat sink, or into electricity by means of an ionic process (taking certain liberties with terminology) through thermopiles or thermionic emitters, or through a photoelectric cell.

In the chemical thermal rocket the transfer of energy to the propellant takes place simultaneously with the energy transformation into heat, that is, in the combustion process (fig. 32). In the nuclear thermal rocket a heat exchanger, that is, the reactor, is involved. Or to introduce a third method, an electric discharge within the propellant could be used to heat the propellant. Such a device is considered, but currently, the situation in regard to it is not clear. For the electric energy, the transfer takes place by means of an electric or magnetic field.

Acceleration of the propellant in a thermal rocket takes place through an expansion nozzle (fig. 33) as discussed previously. The turbojet and ramjet are not used in space propulsion, but are listed here as a matter of general interest. In the electric rocket, the transfer of energy and the acceleration of the propellant take place simultaneously in the subjection of the ion or plasma propellant to the electrostatic or electromagnetic field.

Figures 30 to 33 are summarized in figure 34 to show that the powerplant can be outlined in an orderly manner to bring out essential differences among the different types.

Thermal Rockets

We will next examine the powerplant types in more detail. Figure 35 shows diagrammatically a chemical rocket using fuel and oxidant in solid form, a chemical rocket using fuel and oxidant in liquid form, and a nuclear rocket using propellant (hydrogen) in liquid form. These rockets are termed temperature limited because in each case a temperature limitation, mentioned previously, is imposed.

In the case of the solid chemical rocket, burning takes place from the hollow center of the solid charge toward the walls of the propellant case. Consequently, the case is not subjected to the hot combustion gases until near the end of the combustion period. The expansion nozzle on the other hand is subject to a total temperature equivalent to the combustion temperature for the full burning period. Furthermore, the nozzle is not cooled. With the chemical liquid rocket the fuel and oxidant are pumped from the fuel and oxidant tanks (not shown) by means of the turbopumps indicated or by pressurization to the combustion chamber. Before entering the combustion chamber (combustor) either fuel or oxidant is passed over the combustor and expansion nozzle walls and provides cooling. Both combustion chamber and expansion nozzle walls are subject to the total combustion chamber, but the coolant maintains the walls at a much lower temperature. In each of these rockets the heat flow is from the burning gases to the solid walls and in each case, the gas, therefore, is at the higher temperature.

With the nuclear thermal rocket employing a solid-core reactor to heat the propellant, the heat flow is from the solid fuel elements of the reactor to the gaseous propellant, and the solid material must, therefore, withstand a higher temperature than the propellant.

The essential data that determine the propellant velocity for these rockets are shown in table III, in which the propellant temperature, molecular weight and velocity (listed as jet velocity) as well as material temperatures are summarized. The first column lists the fuels and oxidants. The symbol RP-1 is for a hydrocarbon fuel that is essentially kerosene. The propellant (combustion) temperatures listed are approximate and reasonable changes in them will not change the picture presented. For the solid chemical rocket two material temperatures are specified. Since the nozzle temperature (3200° F) is extremely severe, the duration of operation of solid chemical rockets is limited. Because the liquid chemical rocket combustion chamber and expansion nozzle are cooled by either fuel or oxidant, the duration of operation is not limited by the materials and is adjusted to that suitable for the mission. Operation times for liquid rockets are as much as several minutes. An additional advantage of liquid rockets is that they can be stopped and restarted.

With the nuclear rocket, since heat flow is from the reactor to the propellant, the propellant temperature is limited to a value less than that for the reactor material. A difference of 250° F is shown. Two temperatures are listed without regard to the methods or possibilities of obtaining these temperatures.

Representative propellant molecular weights are shown in the fourth column. Remembering that the propellant velocity varies as the square root of the ratio of propellant temperature to propellant molecular weight (eq. (12)), it is seen that the major cause of variation in propellant velocity (column 5) results from the variation in molecular weight. The improvement in specific impulse, as represented by propellant velocity, is clearly evident as one reads down the figure.

A serious problem with the nuclear thermal rocket is presented by the weight of shielding required to protect the payload against nuclear radiation. In figure 35 a light shield is indicated between the reactor chamber and the propellant pump. The temperature limitation imposed by the solid nuclear reactor could be much less if the nuclear reaction could take place directly within the propellant gas (fig. 36). The materials temperature limit of 7000° F is about the highest that can currently be visualized (fig. 27), although not currently obtainable by any known means, for the solid reactor. The specific impulse for the gaseous reactor is shown to values of 8000 pounds of thrust per pound of propellant per second. Much research will be required before the gaseous reactor can be evaluated in regard to practicability.

Earlier mention was made in the discussion of the effect of ratio of gas pressure in the combustion or heating chamber to the pressure (ambient) into which the propellant discharge take place. At sea level (on the earth's surface) the discharge pressure is that of the atmosphere, 14.7 pounds per square inch. The chamber pressure is generally limited to values of 500 to 1000 pounds per square inch so that excessive wall thickness will not be required. In this case the pressure ratio is between 34 and 68. As the altitude decreases the ambient pressure decreases becoming essentially zero outside the earth's atmosphere. Under this condition the pressure ratio would be infinite. There is a practical limit to the ratio of the discharge nozzle exit area to that of the throat (A_c/A_t), the narrowest section of the nozzle (fig. 35) that prevents realization of the full gain of high pressure ratios. In practice, this ratio is limited to say, 50, which is the correct value for a pressure ratio of about 500. Furthermore, nozzles operating at pressure ratios less than that appropriate for the nozzle area ratio, suffer a loss in efficiency. For this reason, the area ratio up to 50 is chosen depending on the altitude range over which the rocket is to operate. The effects of altitude (pressure ratio) on the specific impulse for two area ratios is shown in figure 37. The lower ratio is designed for a rocket that is to operate in the lower atmosphere. It reaches its maximum efficiency at a pressure ratio of about 100. With a chamber pressure of 500 pounds per square inch, this would be at an altitude of 100,000 feet. The higher area ratio corresponds to use in space and is less efficient at the low altitudes.

Rocket Staging

We have now discussed the effect of chemical propellant choice on specific impulse. In those cases for which obtainable specific impulses are too low for mission accomplishment, rocket staging is used. In figure 38, equation (7) is repeated and certain results are shown for the effect of the ratio of propellant mass (M_P) to gross vehicle mass (M_G) on the attained increase in vehicle velocity. It is assumed that launching is from the earth's surface and that the propellant velocity is 6000 miles per hour (specific impulse, 273 lb/(lb propellant)(sec)). Listed on the curve are the velocity increases required for certain missions. The maximum velocity for a given rocket will be attained with no payload, $\left(\text{that is, } \frac{M_P}{M_G} = \frac{M_P}{M_{St} + M_G}\right)$. Assuming that the minimum value for M_{St}/M_G with no payload is 0.125 to 0.075, $\left(\text{that is } \frac{M_P}{M_{St} + M_P} = 0.875 \text{ to } 0.925\right)$, the maximum attainable vehicle velocity is from 10,000 to 14,000 miles per hour, sufficient for an IRBM mission but insufficient for space missions. As just stated, this deficiency is overcome by staging, that is, by mounting successively smaller rockets on the first rocket (first stage), with the desired payload mounted on the last stage, that is, the smallest rocket. As each stage completes its operation, the powerplant and other structure for this stage are dropped off. As a result, the mass to be accelerated is continually decreased with a consequent decrease in thrust and rate of propellant discharge required. In a multistage rocket, the payload of each stage is considered to be the sum of the masses of all the subsequent stages plus the final payload that is to perform the desired mission. In this case the velocity of the final payload is the sum of the velocity increases of the individual stages computed according to equations (6) or (7)

$$V_{V_t} = V_{V_1} + V_{V_2} + V_{V_3} \text{ etc.} \tag{15}$$

in which

V_{V_t} total velocity increase

V_{V_1} increase during first stage operation

V_{V_2} increase during second stage operation, and so forth

An example of staged rockets is shown in figure 39 for the left-hand rocket. It is assumed that a 100-pound payload is to be used as a moon probe. It is further assumed that the required velocity increase is to be achieved in three stages of 8000 miles per hour each, further, for each stage the ratio M_P/M_G is 0.82 and the ratio of $\frac{M_{St}}{M_{St} + M_P + M_{PL}}$ is 0.09, leaving a ratio of M_{PL}/M_G of 0.09. These ratios are held constant throughout the calculations for the oxygen - C_8H_{18} rocket. (RP-1 is considered equivalent to C_8H_{18}). Certain assumptions are made in regard to gravity losses ($g_e t$). The third stage of the oxygen - C_8H_{18} rocket is computed to be as shown. The total third stage weight, including payload of 1100 pounds, becomes the payload of the second stage. (Ratios of payload to gross stage weight are, in general, appreciably higher than the value used here; but this does not change the general conclusions as presented.) Following through these calculations, the gross weight of the vehicle at takeoff is 135,500 pounds. The desired velocity increase of 24,000 miles per hour has been achieved, although the ratio of $(M_{St_3} + M_{PL_3})/M_{G_1}$ is (100 + 100)/135,500 or 0.0015, instead of about 0.02 required (fig. 38) if a single stage rocket could have been built to do the job.

A higher specific impulse propellant can be used to decrease the number of stages and the gross weight at launch to place the 100-pound payload in the proximity of the moon, or to permit a greater final payload for the same gross weight at launch. The estimated results for the greater final payload are shown with the middle vehicle in figure 29 for a three-stage rocket system employing fluorine-hydrogen as the propellant. In the computations, because of lower propellant density, the ratio of M_{St}/M_G is assumed to be 0.12. Because of the higher propellant velocity (specific impulse), the payload of the first stage is 30,200 pounds, a two to threefold increase over that with the oxygen - C_8H_{18} system. The increase is continued through the other two stages so that the final payload is 1500 pounds. The use of fluorine as an oxidant presents many problems. For instance, it is very corrosive. For this reason, a third rocket system is presented to show the kind of improvement that might be realized by using the upper two stages with hydrogen-oxygen as a chemical high-energy propellant. For these two upper stages, M_{St}/M_G is again assumed to be 0.12. In this case the final payload is 400 pounds.

The nuclear thermal rocket permits still higher specific impulses. The analysis of the gains to be realized through its use are dependent on the amount of shielding required and on the temperature which the reactor can withstand. Certain gains that may be obtained will be briefly discussed later.

Means of Increasing Temperature Limits

The thermal limitations imposed by the rockets shown in figure 35 can be greatly lessened if rockets of the types shown in figure 40 can be developed. In the first two examples the gas is heated by nuclear fission or fusion, heating taking place directly within the gas under such temperature conditions that a plasma exists, and the hot plasma is prevented from contacting the heating chamber walls by means of an electromagnetic field. The second example represents a variant of the example shown in figure 36. In the third example electric discharge heating is used in which the hot ionized particles do not impinge on the chamber wall. The devices illustrated are simply meant to represent kinds of devices under consideration without regard to their present applicability. Other devices are also being considered, which are very much in the research stage. Considerable research is needed to narrow the field down to those projects that warrant an intensive development effort.

Electric Propulsion Systems

The propulsion system in which the propellant is accelerated by means of an electric or electromagnetic field is shown diagrammatically in figure 41 (see also fig. 7). In this case, the propellant is fed into a mechanism (labelled ionization device) and charged electrically. Assuming the charge is acquired by the removal of one or more electrons per particle, the propellant becomes either an ionized gas in which case the electrons are physically separated from the ions; or it becomes a plasma in which case the conditions are such that the ions and electrons can remain together without recombination. For the ionized gas, an electrostatic field is the accelerator; for a plasma, an electromagnetic field is the accelerator. In either case, propellant velocities of several hundred thousand miles an hour (specific impulses of 10,000 lb/lb propellant discharged per sec or more) can be attained.

Another factor which must now be considered is the power in the propellant jet.

$$P_P = \frac{1}{2} \frac{M_P}{t} V_P^2 \tag{16}$$

in which P_P is power in jet. The force, or thrust, produced by the jet is

$$F = \frac{M_P}{t} V_P$$

in which F is thrust produced. Obviously, this force can be kept constant by decreasing the mass rate of propellant discharge M_P/t and increasing V_P proportionately. However, in this case, the jet power is increased to the same degree the velocity is increased.

With the chemical rockets,the powerplant mass, that is, the mass of propellant pumping system plus the combustion chamber plus the expansion nozzle is quite low. With the nuclear thermal rocket the weight will be reasonably low, providing shielding weights are kept down. With the electric rocket systems, ion or plasma jets, this is not the case. Representative values are shown in table IV. The second column lists the horsepower per pound of thrust. This value, as pointed out previously, varies directly with specific impulse (propellant velocity), listed in the first column. The pounds of powerplant per pound of thrust shown for the nuclear electric rocket in the last column indicate a major problem to be overcome or compensated for. As mentioned previously, with the electric nuclear rocket (ion jet or plasma jet), the acceleration results from the action of an electric or electromagnetic field. The power in the propellant jet must be supplied from this field and must be generated by some form of electric generating system. Also, as mentioned previously, the energy source for generating this power will almost certainly be nuclear. Currently, this means a thermal turboelectric generator with the working fluid heated in the reactor (fig. 42). Since the system is acting in space with no external cooling medium available, the radiator must lose heat through radiation. This will mean a relatively large radiator. In computing the mass per pound of thrust for the electric rocket, a minimum weight of ten pounds per horsepower has been assumed for the electric power system shown in figure 41. Using this figure and assuming a power-plant weight of 40 percent of the gross vehicle mass, the ratio of thrust to gross mass is 10^{-4}. For this reason, the electric propulsion system can be considered only for those flight conditions in which the resultant of all gravitational forces acting on the vehicle counter to the desired direction of motion is less than 10^{-4} of the vehicle mass, or that a velocity has already been given to the vehicle to counterbalance the effect of these gravitational forces. In any case, the vehicle acceleration will be extremely low, of the order of 10^{-4} times that due to gravity at the earth's surface.

With the electric propulsion system, it must be remembered that the electric power must accomplish the ionization of the propellant as well as the acceleration. Referring again to figure 21, going through the necessary calculations will show that at the specific impulse given in table IV with cesium as the propellant in the electric rocket the ionization energy (or power) is about 0.02 percent of the propellant jet energy (or power). With hydrogen, the figure is 15 to 20 percent, an appreciable loss.

A partial comparison of thermal and electric propulsion systems is shown in table I in which a trip from earth orbit to a moon landing and return to earth orbit is considered. The influence of propellant velocity on propellant mass ratio (M_P/M_G) and on the ratio of propellant jet power to thrust produced (P/F) is shown. As discussed previously, chemical rockets for this mission would have to be staged, since a maximum ratio of M_P/M_G of 0.85 to 0.90 is about as high as is practical in a single stage.

Electric Power Generation

Since electric power will, in general, be required aboard spacecraft aside for uses other than propulsion additional means of generating electric power for spacecraft are under consideration. Figure 43 is presented to show a choice of power generation system based on required power output and required use time. The hydrogen-oxygen fuel cell is a form of chemical battery. Two additional types of systems are also being considered - thermionic emitters and thermopiles (fig. 44). Based on current results, their use will be appropriate for the longer operating times, say, of 10 days or more and power outputs up to the low kilowatt range. Much research is being conducted on these systems to improve their efficiencies which are currently of the order of 5 to 8 percent. Either a nuclear reactor, a radioisotope decay source, or the sun can be used for the energy supply.

Current Status of Propulsion Systems

The general development status of the various space propulsion systems is summarized in figure 45, which is from Dr. Sutton's 1959 Minta Martin lecture (listed in the bibliography). The uses of certain of the systems are summarized in figure 36, also from Dr. Sutton's paper.

Other Types of Propulsion Systems

Propulsion systems other than those already discussed are under consideration. The solar sail using the force of the sun's radiant energy has been mentioned (fig. 47). The force varies inversely as the square of the distance from the sun. At the distance to the earth, the force is 2×10^{-7} pound per square foot. This means for a thrust-mass ratio of 10^{-4}, that is, 500 square feet of sail is required for each pound of vehicle mass.

Another method that has been discussed is to use a series of nuclear explosions (fig. 48), the explosive force of which would accelerate the vehicle. There is the matter of using nuclear fission products directly as the propellant (fig. 49). Again, we might consider photons (energy impulses) ejected from a high temperature surface (fig. 50), in which the necessary heat is supplied by a nuclear reactor. Systems such as these are being studied to determine the interest that should be placed in them and to determine the research necessary for practical developments.

APPLICATIONS

In general, as the required velocity change is increased or as the mass of the payload is increased, the higher specific impulse nuclear powered propulsion systems become more advantageous. The reason for this is, of course, the higher specific impulse, gives a higher vehicle velocity change for a given ratio of propellant mass (M_P) to vehicle gross mass (M_G). In addition, with nuclear energy, greater payloads mean that the nuclear reactor and shielding masses become a smaller portion of the total mass, since the specific mass of the reactor plus shield, that is, pounds per horsepower output, decreases as the power increases.

In the case of the electrical systems, the mass of the powerplant is still so great, that the system can only be used at extremely low ratios of thrust to vehicle mass.

In figures 51 and 52 respective estimates are presented of the gross mass required in an earth orbit to permit a manned mission to land on the moon or Mars and return to earth orbit.

All weights shown in the bar graphs are the initial weights that must be launched into an orbit at a 400 mile altitude to get the mission underway. For these manned missions, an auxiliary chemical rocket vehicle is carried along to land part of the crew, with exploration equipment, on the surface of the moon or Mars and to take them back to the mother ship, which remains in an orbit having an altitude of about 100 miles above the moon or 200 miles above Mars. The weight of this auxiliary rocket, plus its fuel and the crew's exploration equipment, is labeled "Landing and Exploration" on the bar graphs. The "Basic Payload" consists of the crew, cabin, environment control equipment, navigation and communication equipment, and scientific instrumentation, but does not include the subsistence supplies, since these disappear during the course of the mission.

In each figure, "I" represents the specific impulse in pounds of thrust per pound of propellant discharged; F/W_o, the ratio of thrust produced to vehicle mass; and "a" the ratio of powerplant mass to propellant jet horsepower. For the moon mission the higher specific impulse chemical rocket gives results comparable to that for the nuclear (thermal) rocket. The nuclear electric rocket given somewhat lower gross weights, but the low thrust to mass ratio would require of the order of 50 days to leave the earth orbit, too long a time for a moon trip that could be completed in a few days.

For the Mars trip the nuclear systems show great improvement over the chemical systems. The higher gross mass involved and the higher velocity changes required both work in favor of the nuclear systems. Here the choice between the nuclear thermal rocket (listed as Nuclear Rocket) and the nuclear-electric propulsion system is not clear. More research is needed.

A simpler mission, a Mars probe from the earth is illustrated in figure 53. Radiation shielding requirements with the nuclear thermal rockets are lessened, because living matter is not aboard. The nuclear rockets result in considerable improvement in payload. Takeoff from the earth with a nuclear thermal rocket (listed as nuclear boost) entails radiation hazards that cannot, in general, be tolerated at the present state of development.

CONCLUSIONS

For flight outside the earth's atmosphere the chemical thermal rocket currently dominates the picture and will do so until the research on nuclear devices leads to practical engineering applications. The choice between liquid or solid chemical propellants for space propulsion is currently being decided in favor of the former, except for smaller upper stages. Additional research is needed to determine the full potential of either system.

As progress is made in adapting nuclear energy to spacecraft, the currently too difficult space missions will become practical. The choice between nuclear-thermal and nuclear-electric systems is uncertain at this time. Again, research is required.

The research emphasis in flight propulsion from air-breathing to rocket engines has increased the required research areas many times. There is a great need for the kind of exploratory research that will permit us to focus our efforts. The expense and time involved makes coordination and organization of effort increasingly important.

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Table I. - REQUIRED VELOCITY CHANGES FOR TRIP FROM EARTH TO MOON AND RETURN

(1) TO REACH EARTH ORBIT	+ 17,000 MPH
(2) TO LEAVE EARTH ORBIT	+ 7,000 MPH
(3) TO ENTER MOON ORBIT	- 1,500 MPH
(4) TO LAND ON MOON	- 4,000 MPH
(5) TO REENTER MOON ORBIT	+ 4,000 MPH
(6) TO LEAVE MOON ORBIT	+ 1,500 MPH
(7) TO ENTER EARTH ORBIT	- 7,000 MPH
(8) TO LAND ON EARTH	- 17,000 MPH
TOTAL VELOCITY CHANGE	59,000 MPH

TABLE II

$\frac{M_V + M_P}{M_V}$	$\ln \frac{M_V + M_P}{M_V}$
2.5	0.92
5.0	1.61
10.0	2.30
20.0	3.00

Table III. - COMPARISON OF TEMPERATURE LIMITED THERMAL ROCKETS

PROPELLANT	TEMPERATURES, °F PROPELLANT	TEMPERATURES, °F MATERIAL	EFFECTIVE MOL. WT.	JET VELOCITY, MPH
		CHEMICAL		
SOLID	5500	600[1] / 3500[2]	30	5000
RP-1 - O_2	5500	1200	22	6000
H_2-F_2	5000 / 7000	1400	11	8500
		NUCLEAR		
H_2	3250	3500	2	14,500
	5250	5500		18,500

[1] CASE
[2] NOZZLE

Table IV. - SPECIFIC PERFORMANCE OF SPACECRAFT POWERPLANTS

	SPECIFIC IMPULSE, LB THRUST / (LB PROPELLANT)(SEC)	HP, LB THRUST	LB ENGINE PER LB THRUST
CHEMICAL ROCKET	410	12	0.020
NUCLEAR THERMAL ROCKET	1,180	35	0.060
NUCLEAR ELECTRIC ROCKET	13,600	400	4,000 TO 10,000

TABLE V. - COMPARISON OF ROCKET PROPELLANT REQUIREMENTS FOR MOON LANDING AND RETURN.

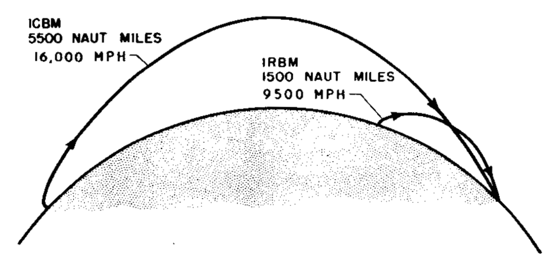
$^{*}\Delta V_V = 25{,}000$ MPH

ROCKET	PROPELLANT	$V_{PROP.}$, MPH	$\frac{M_P}{M_G}$	$\frac{HP}{F, LB}$
		THERMAL ROCKETS		
CHEMICAL	$RP\text{-}1\text{-}O_2$	6,500	.98	8.6
CHEMICAL	$H_2\text{-}F_2$	9,000	.93	12.3
NUCLEAR	H_2	26,000	.60	35.0
		ELECTRO-MAGNETIC ROCKETS		
ION	Cs^+	300,000	.09	409.0

$$V_V = V_{PROP.} \ln \frac{M_G}{M_G - M_P}$$

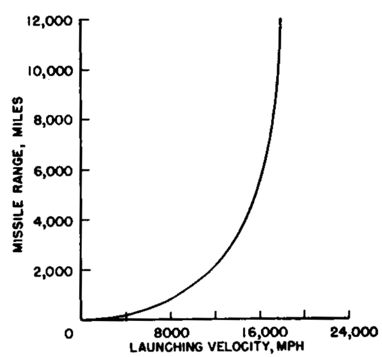
$^{*}\Delta V$ REQUIRED FROM EARTH SATELLITE TO MOON LANDING AND RETURN TO EARTH SATELLITE

15493-B



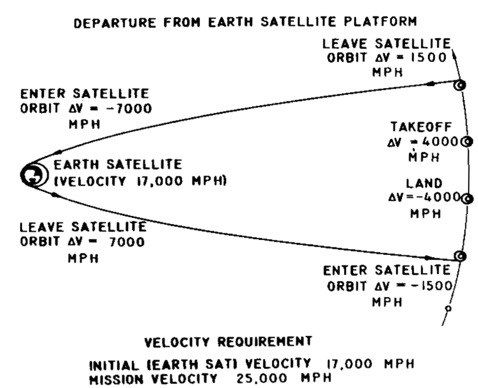
15491

Figure 1. - IRBM and ICBM trajectories.



15811

Figure 2. - Approximate variation of missile range with launching velocity.



15492-A

Figure 3. - Moon landing and return.

15214

B

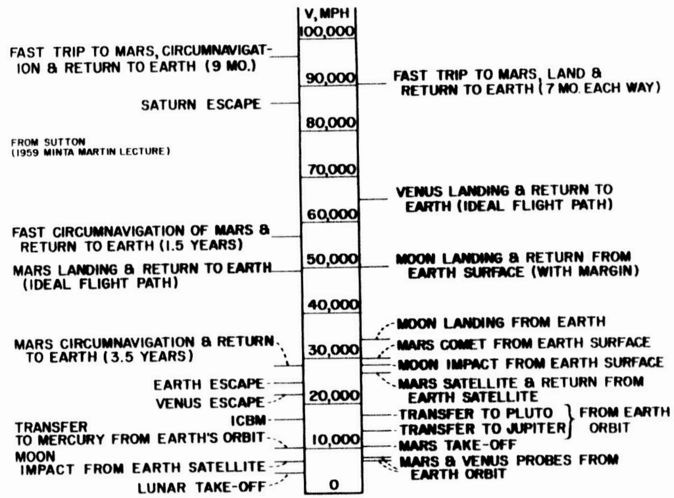


Figure 4. - Required velocity changes for various space missions.

ENERGY SOURCE	CHEMICAL NUCLEAR
PROPELLANT	SOLID LIQUID GAS PHOTONS
POWER PLANT	TRANSFORM ENERGY TO HEAT OR ELECTRICITY TRANSFER ENERGY TO PROPELLANT ACCELERATE PROPELLANT
MATERIALS	METALS CERAMICS PLASTICS LUBES
HEAT TRANSFER FLUID	GAS LIQUID

Figure 5. - Propulsion system components for aircraft or spacecraft.

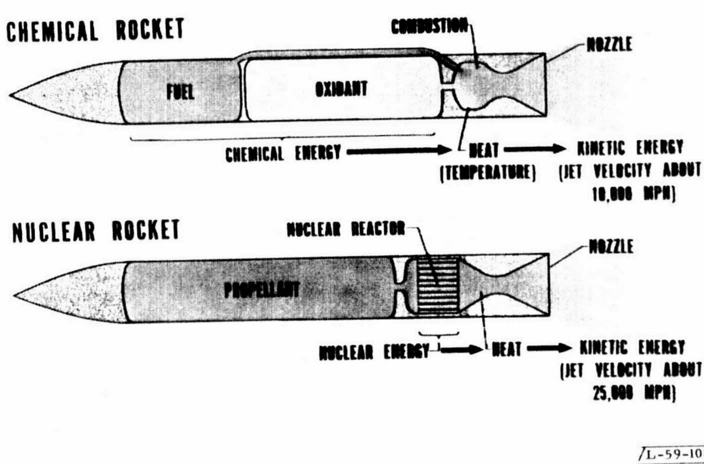


Figure 6. - Thermal rockets.

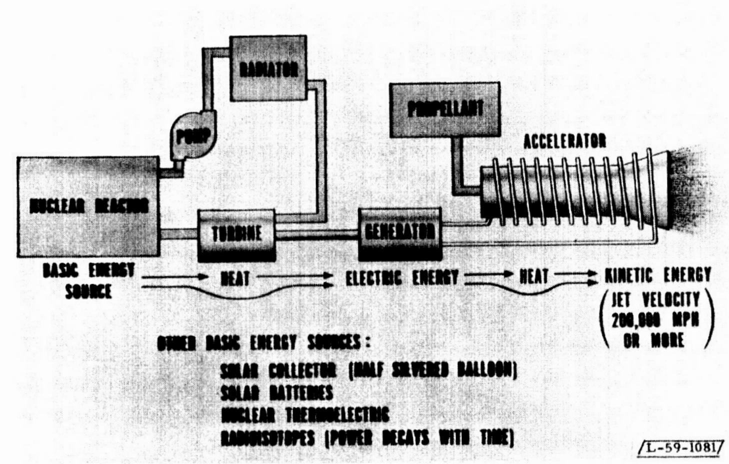


Figure 7. - Electric rocket.

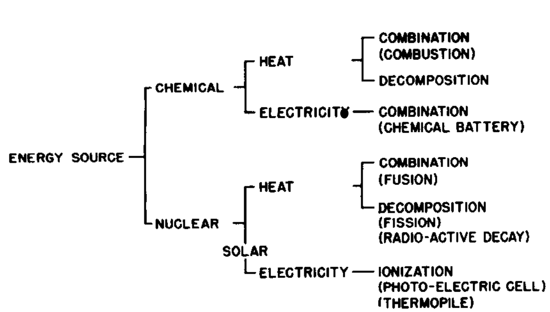


Figure 8. - Energy conversion to heat or electricity.

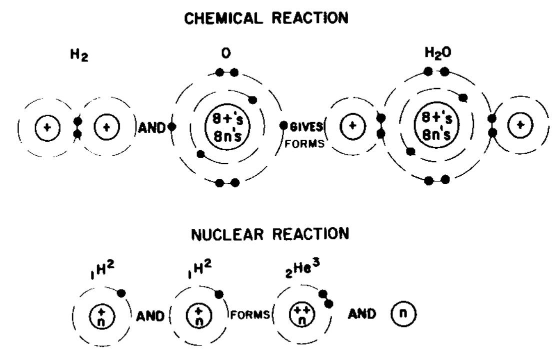


Figure 9. - Chemical and nuclear reactions.

CHEMICAL -

$$H_2 + \frac{1}{2}O_2 \rightarrow H_2O + eV$$

$$H_2O_2 \rightarrow H_2O + \frac{1}{2}O_2 + eV$$

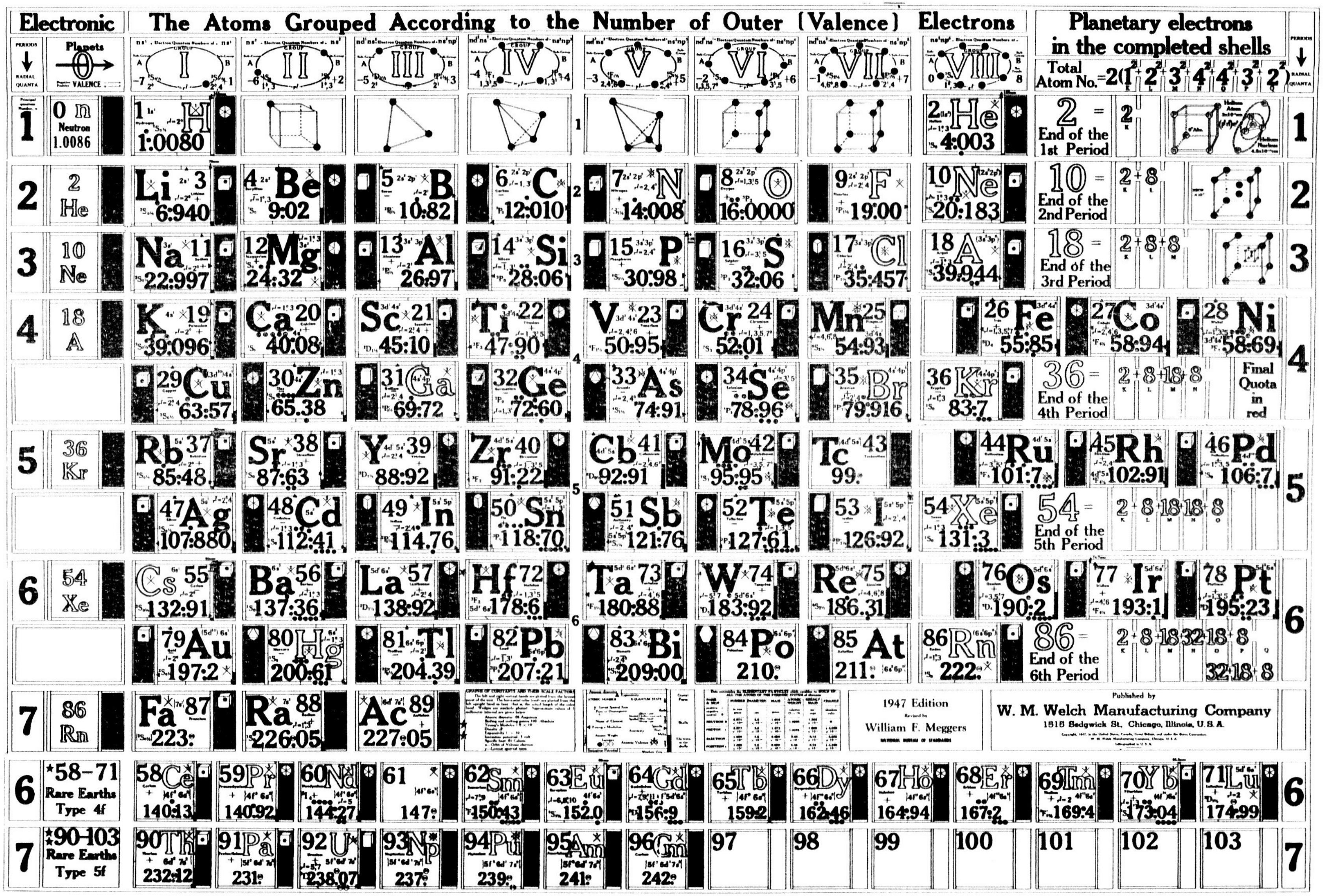
NUCLEAR -

$$_1H^2 + {_1H^2} \rightarrow {_2He^3} + n + eV$$

$$_{92}U^{235} \rightarrow {_{55}Cs^{137}} + {_{56}Ba^{140}} + {_{38}St^{90}} + \cdots + eV$$

$$_{84}Po^{210} \rightarrow {_{82}Pb^{206}} + He + eV$$

Figure 10. - Typical chemical and nuclear reactions.



CS-15784

Figure 11. - Periodic chart of the atoms.

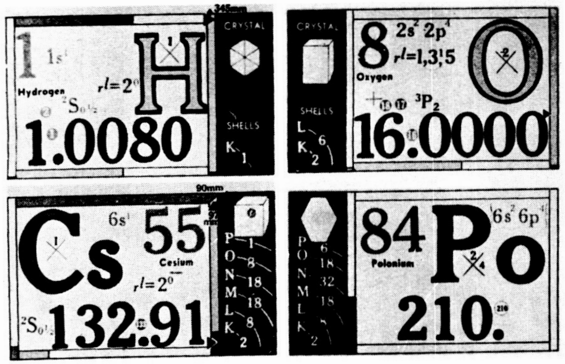


Figure 12. - Enlargements from figure 10.

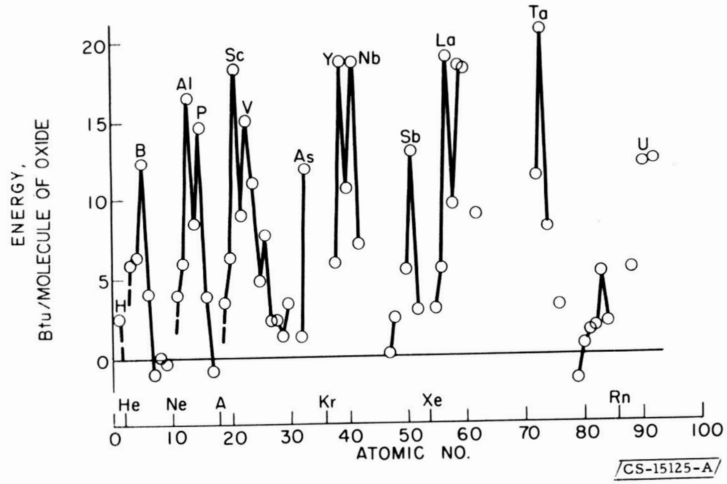


Figure 13. - Reaction energy with oxygen.

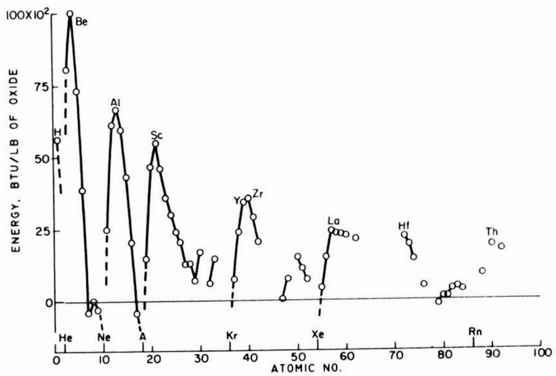


Figure 14. - Reaction energy with oxygen.

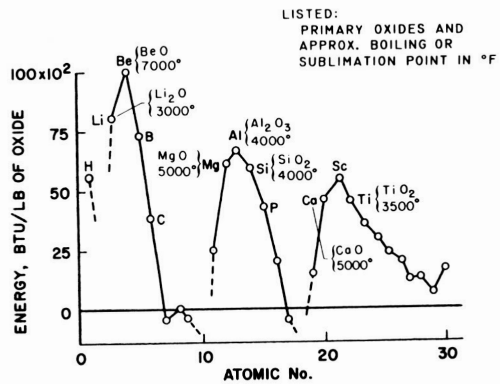
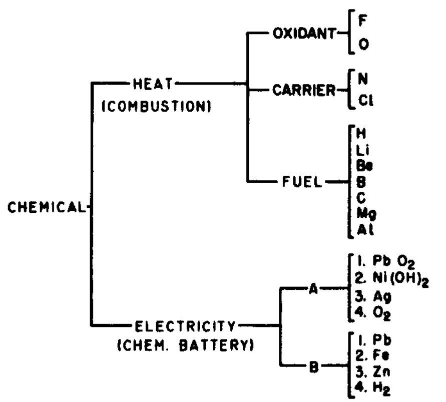


Figure 15. - Detail of reaction energy with oxygen.



15129

Figure 16. - Chemical elements of interest as rocket fuels or oxidants.

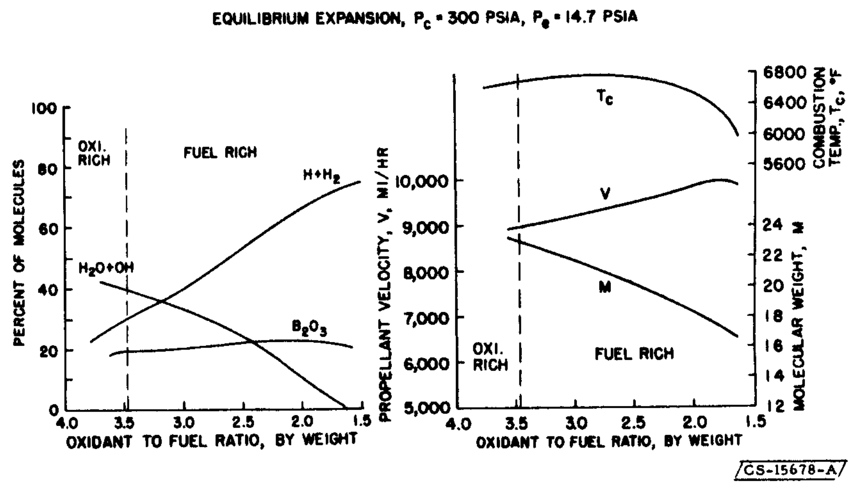
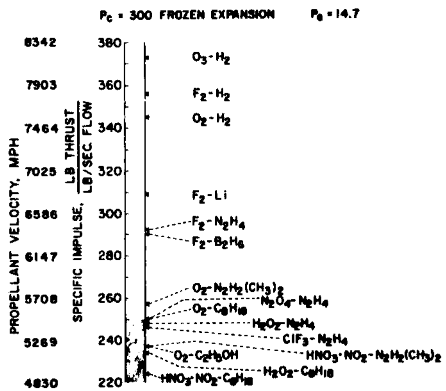
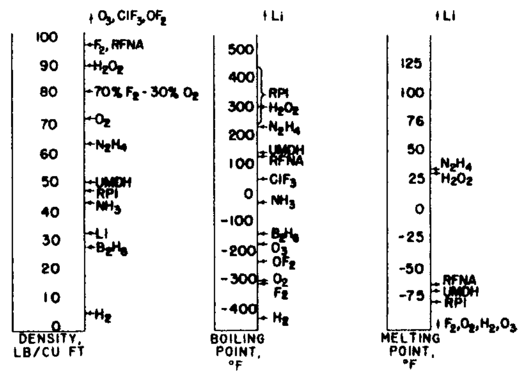


Figure 17. - Theoretical performance of O_2-B_2H_6.



15682-A

Figure 18. - Performance of representative rocket propellants.



15153

Figure 19. - Physical properties of fuels and oxidants.

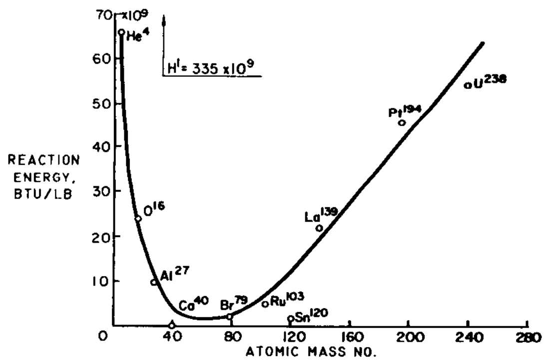


Figure 20. - Maximum nuclear energy available from fission or fusion reactions.

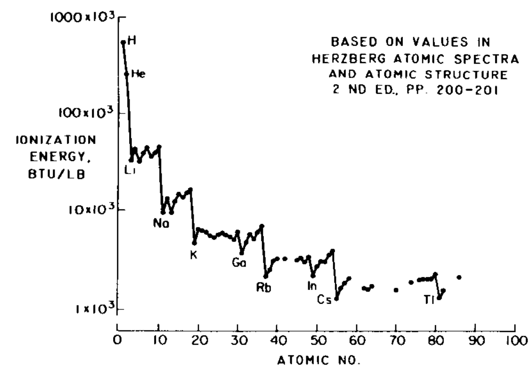


Figure 21. - Ionization energies of the elements in BTU per lb (first electron).

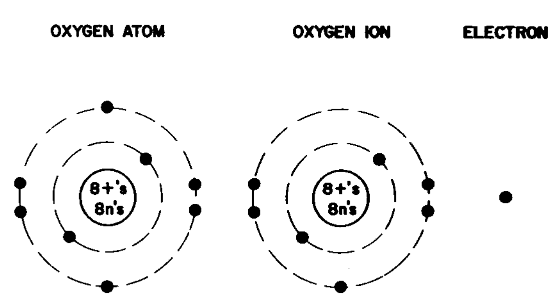


Figure 22. - Ionization process.

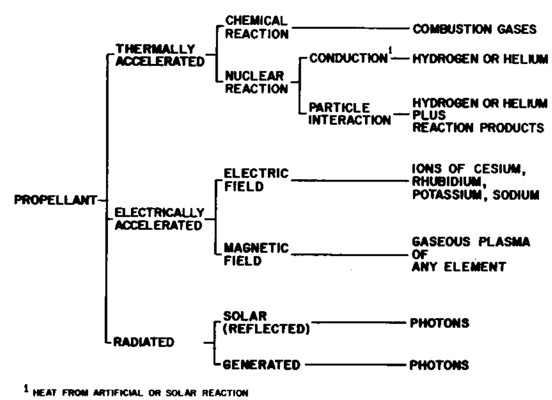


Figure 23. - Propellants.

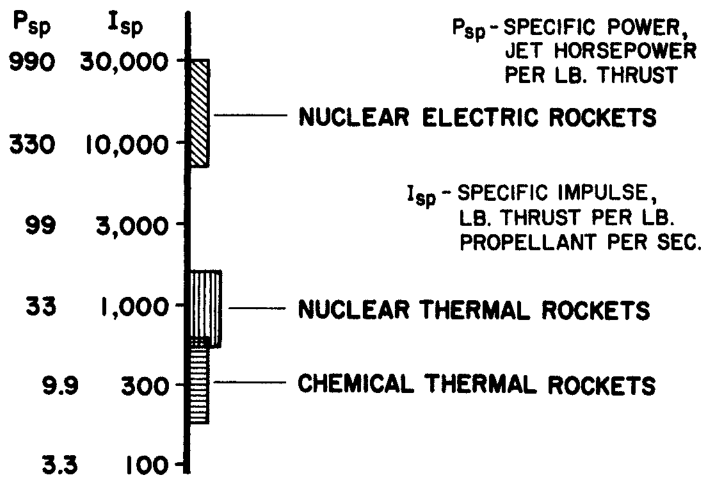


Figure 24. - Propulsion system performance.

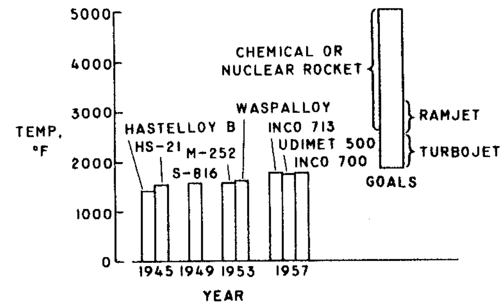


Figure 25. - Operating temperature of aircraft turbine materials.

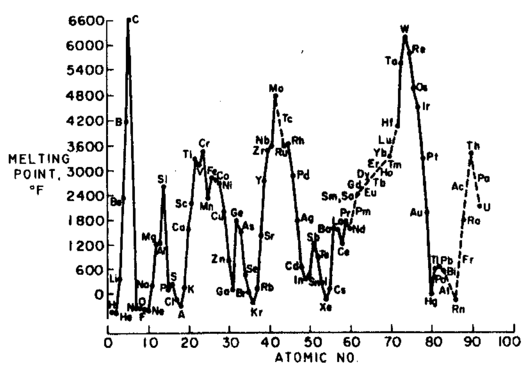


Figure 26. - Melting temperatures of the elements vs atomic number.

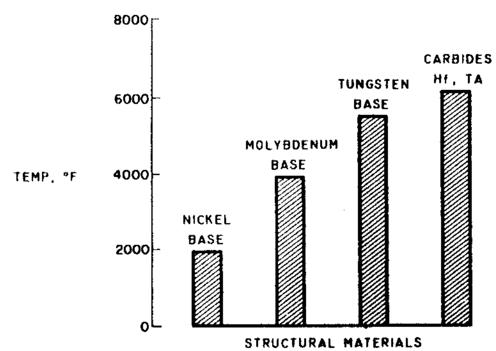
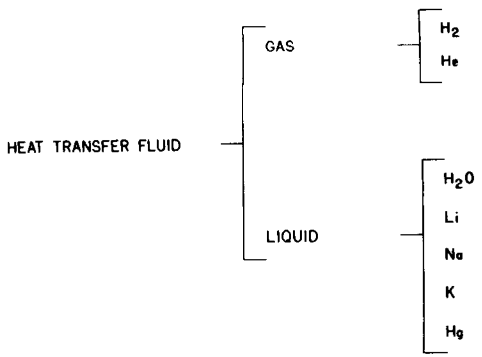
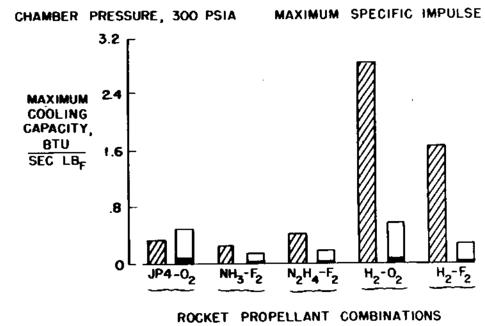


Figure 27. - Estimated operating temperatures for materials.



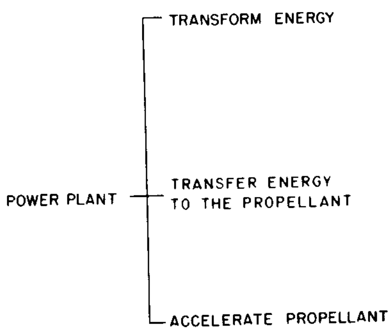
15168-A

Figure 28. - Heat transfer fluids.



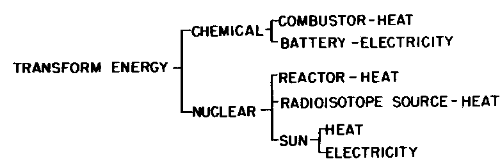
14819

Figure 29. - Cooling capacities of propellants.



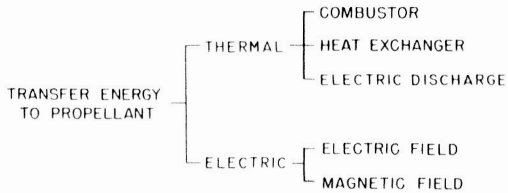
15137-B

Figure 30. - Power plant operations.



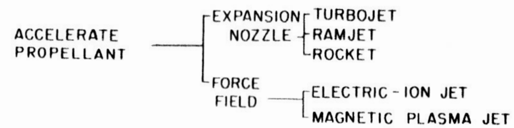
15138

Figure 31. - Energy transformation.



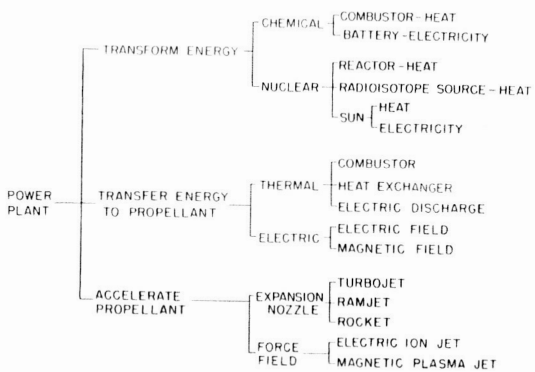
15140-A

Figure 32. - Energy transfer.



15139

Figure 33. - Propellant acceleration.



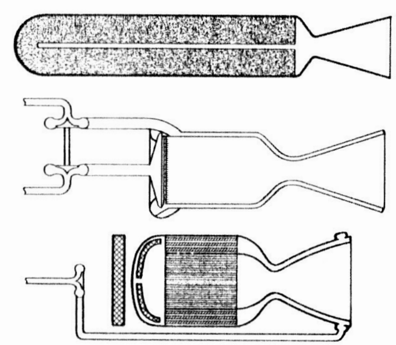
15135-A

Figure 34. - Power plant breakdown.

CHEMICAL SOLID

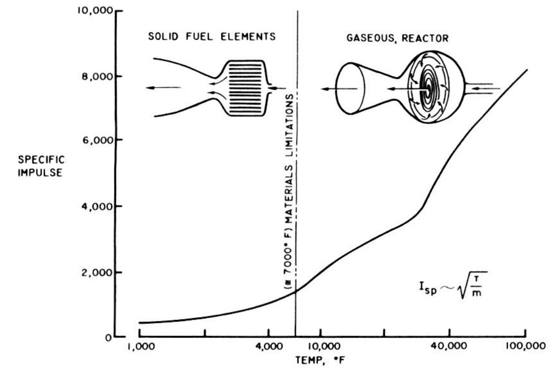
CHEMICAL LIQUID

NUCLEAR



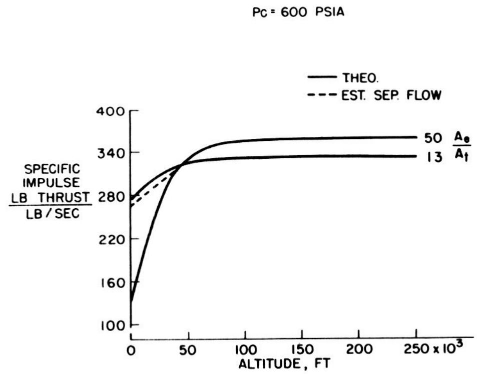
15193-A

Figure 35. - Temperature limited thermal rocket engines.



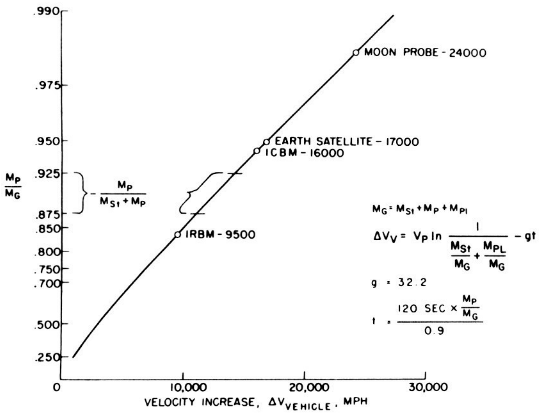
16271-A

Figure 36. - Nuclear rocket propulsion.



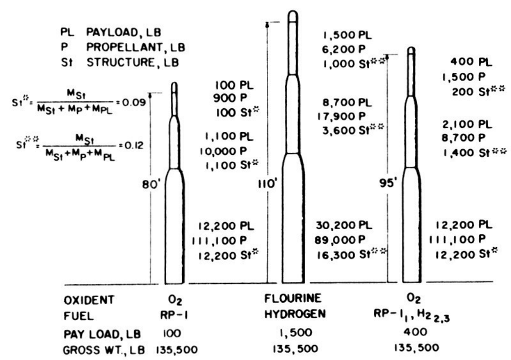
14956

Figure 37. - Performance of O_2-RPI at altitude.



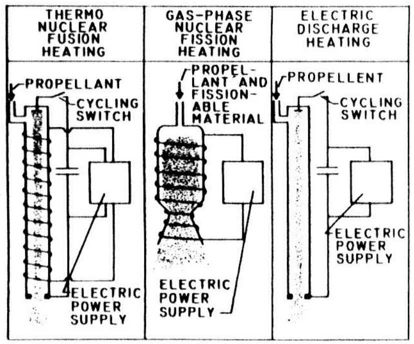
16463

Figure 38.



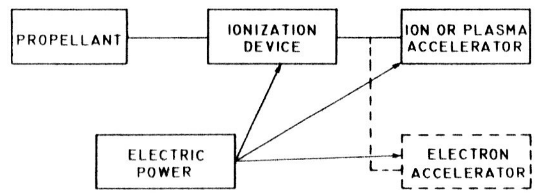
16739 B

Figure 39. - Rocket systems for moon probe.



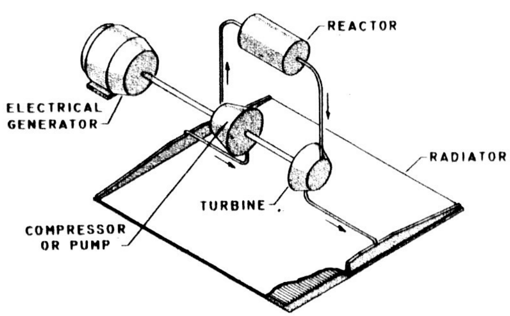
15758

Figure 40. - Thermal rockets using magnetically contained plasma as propellant.



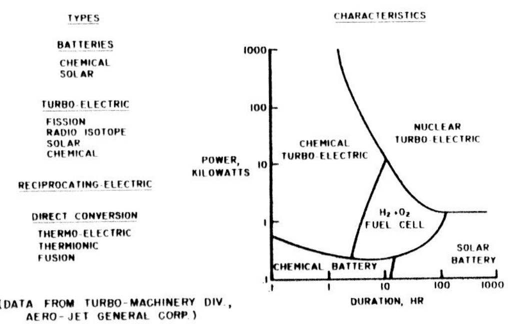
15330

Figure 41. - Components of ion or plasma electro-magnetic rocket.



14579

Figure 42. - Simplified cycle arrangement.



16265

B

Figure 43. - Electrical power generation.

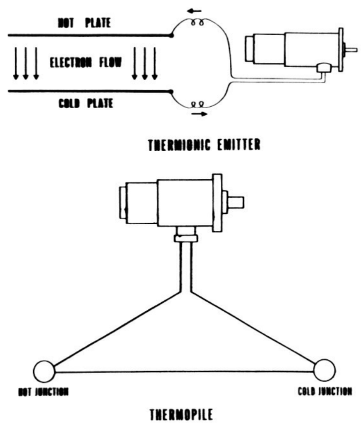


Figure 44. - Spacecraft power generation.

17469

	ANALYSIS	BASIC RESEARCH	ENGINE DEVELOPMENT FACILITIES	DEVELOPMENT OF TERRESTIAL VERSION	PILOT PRODUCTION OF TERRESTIAL VERSION	DEVELOPMENT OF SPACE VERSION
LIQUID PROPELLANT	○	○	○	○	○	○
SOLID PROPELLANT	○	○	○	○	○	○
MONOPROPELLANT	○	○	○	○	●	●
FISSION	○	▨	▨	▨	●	●
FUSION	▨	▨	●	●	●	●
RADIOACTIVE ISOTOPE DECAY	▨	▨	▨	●	●	●
ARC HEATING	▨	▨	▨	●	●	●
MAGNETO-PLASMA	○	▨	●	●	●	●
ION	○	▨	▨	●	●	●
SOLAR HEATING	○	▨	▨	▨	●	●
SOLAR SAIL	▨	●	●	●	●	●

FROM SUTTON (1959 MINTA MARTIN LECTURE)

○ COMPLETED, AVAILABLE
▨ UNDER WAY
● NOT ACCOMPLISHED OR AVAILABLE

Development Status of Various Rocket Engines

Figure 45. - Development status of various rocket engines.

MANEUVER	CHEMICAL	FISSION	ISOTOPE DECAY	ARC HEATING	MAGNETO-PLASMA	ION	SOLAR HEATING	SOLAR SAIL
ESCAPE FROM ORBIT	○	○	▨	▨	▨	▨	▨	▨
INTERPLANETARY TRANSFER	▨	▨	○	○	○	○	○	▨*
CHANGE OF PLANE	▨	▨	○	○	○	○	○	▨*
ORIENTATION	○	▨	○	○	○	○	○	○
TRAJECTORY CORRECTIONS	○	▨	○	○	○	○	○	●
TARGET APPROACH	▨	▨	○	○	○	○	○	●*
CAPTURE	○	○	▨	▨	▨	▨	▨	●

○ SUITABLE
▨ ACCEPTABLE
● UNDESIRABLE
* ONE-WAY ONLY

Suitability of Various Propulsion Systems to Different Interplanetary Maneuvers

FROM SUTTON (1959 MINTA MARTIN LECTURE)

Figure 46. - Suitability of various propulsion systems to different interplanetary maneuvers.

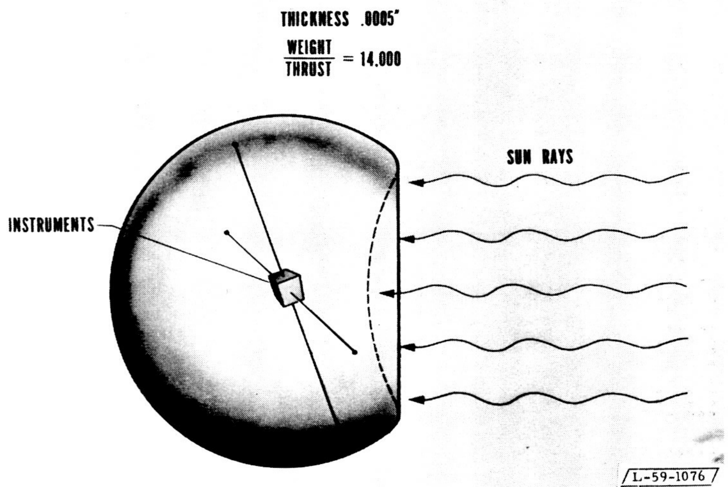
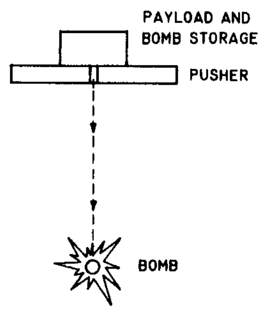
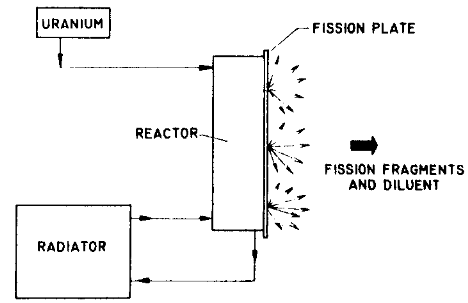


Figure 47. - Photon sail.



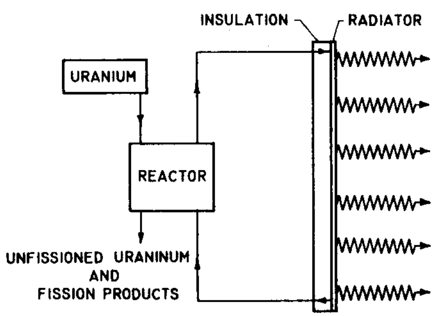
17325

Figure 48. - Bomb rocket.



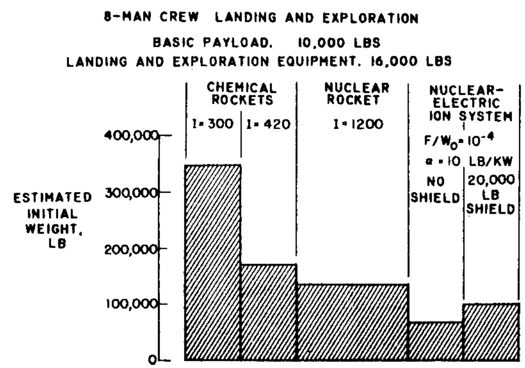
17262

Figure 49. - Fission product propulsion system.



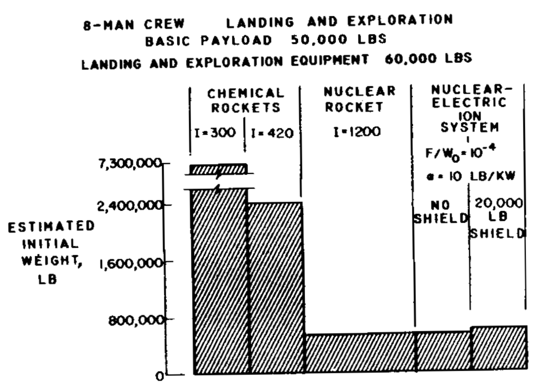
17326

Figure 50. - Photon rocket.



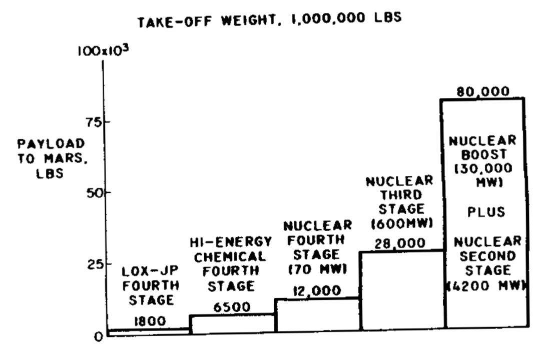
15306-A

Figure 51. - Round-trip to moon.



15307-A

Figure 52. - Round-trip mars expedition.



15505

Figure 53. - Mars payload capability.